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ADP010775

TITLE: Tutorial on Repair Software

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TITLE: Aging Aircraft Fleets: Structural and Other Subsystem Aspects [le Vieillissement des flottes d'avions militaires : aspects structures et autres sous-systemes]

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TUTORIAL ON REPAIR SOFTWARE

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1.0 INTRODUCTION

Throughout the world military and commercial aircraft fleet are being used beyond their original design life. This is primarily due to the reduction in the budget for procurement of new systems and ever increasing cost of acquiring new aircraft. This has resulted in paying more attention to enhancing life of aircraft structures and at the same time maintaining the safety of flight. Improved life enhancement techniques and repair concept are being developed to keep maintenance cost low, reduce down time of aircraft for repairs, reduce inspection requirements without jeopardizing the safety of aircraft structures.

To reduce the down time of aircraft for repairs and perform more reliable durability and damage tolerance analyses, a number of software programs have been developed. These software programs are user friendly and a user does not have to be an expert in the repair technology or durability and damage tolerance analyses. For most of these programs basic knowledge of stress analyses, fatigue and fracture mechanics is required. This tutorial discusses some of these programs, and steps involved in the analyses of repairs to assure safety of flight.

2.0 SOFTWARE PROGRAMS FOR REPAIR DESIGN, AND DURABILITY AND DAMAGE TOLERANCE ANALYSES

A number of software programs have been developed for designing repairs for aircraft structures and performing durability and damage tolerance analyses. These programs are operational on a personal computer (PC). Some of these programs are briefly described here.

1) AFGROW

This code has been developed by US Air Force Wright Patterson Air Force Base (WPAFB), Ohio, for durability and damage tolerance analyses of aircraft structures under constant amplitude and spectrum loading. Code has capability to design composite patch repairs for metallic structures. Crack growth life predictions can be made in an aggressive environment accounting for the corrosive effects on crack growth. The code is user friendly with an excellent graphical user interface. The code has a good database of material properties needed for damage tolerance analysis. A user has an option to input own material properties. The code has a built in library of stress intensity factors for a number of crack configurations and structural geometries. The user has an option to input own stress intensity factors. For crack growth predictions under spectrum loading, the user has to input loads spectrum. The code takes spectrum in a certain format. Majority of airframe manufacturers have own ways of generating spectrum. Hence, a translator is required to convert input spectrum in the format used by the code. The translator varies with the input spectrum format. Translators for a number of spectra formats have been included in the code. If a user's spectrum input is not in a format used in AFGROW code, Mr. Jim Harter at WPAFB, Ohio, may be contacted for assistance in developing a translator.

For crack growth predictions under spectrum loading, a number of retardation models have been included in the code. Retardation models included are-
a) Willenborg
b) Wheeler
c) Crack Closure.

Paper presented at the RTO AVT Lecture Series on "Aging Aircraft Fleets: Structural and Other Subsystem Aspects", held in Sofia, Bulgaria, 13-16 November 2000, and published in RTO EN-015.

The retardation parameters needed for these retardation models are included in the code for some of the materials. A user has option to input own retardation parameters.

The code has capability for predicting fatigue life under spectrum loading. Strain-life approach has been used for fatigue life predictions. The strain life approach requires cyclic stress-strain and strain-life data for the structural material of interest. The cyclic stress-strain and strain-life data for some materials have been included in the code.

This code has capability to design composite patch repairs. A knowledge-based system to design repairs has been developed in the code. The code recommends the most suitable material for composite patch design based on the following considerations-

a) Thickness to be repaired.
b) Loads spectrum experienced by the aircraft component.
c) Stress level in the spectrum

Repair material choices available are- boron/epoxy, graphite/epoxy, and GLARE. The properties of these materials are included in the code. The code recommends ply orientations and thickness for the composite patch. The code uses damage tolerance approach for designing repair patches. The design of repair patches is based on using ductile adhesive FM-73 for bonding process.

2) NASGRO

This user-friendly program has been developed by NASA Johnson Space Center and is available in public domain. The program is primarily for damage tolerance analyses of structures. The program has an excellent database for material properties varying from aluminum to steel, plate, sheet, forging, etc.

The program uses boundary element technique to compute stress intensity factors. Crack growth predictions can be made under constant amplitude and spectrum loading. The program does not have capability to design repairs. However, it is a very useful tool for the damage tolerance analysis of structures and repairs.

3) RAPID

This program has been developed under FAA sponsorship with support from US air Force and is primarily for mechanically fastened repairs of transport aircraft (Reference 1). The program has an excellent Graphical User Interface (GUI). The program is available in public domain. The program is not suitable for damage tolerance analysis of aircraft structures. The program has capability to perform repair analysis under constant amplitude as well as spectrum loading.

The program has an excellent database for material and fastener properties. The program is suitable for designing three different types of repairs-

1) One external and one internal doubler (Figure 1).
2) Two external doublers (Figure 2).
3) One external doubler (Figure 3)

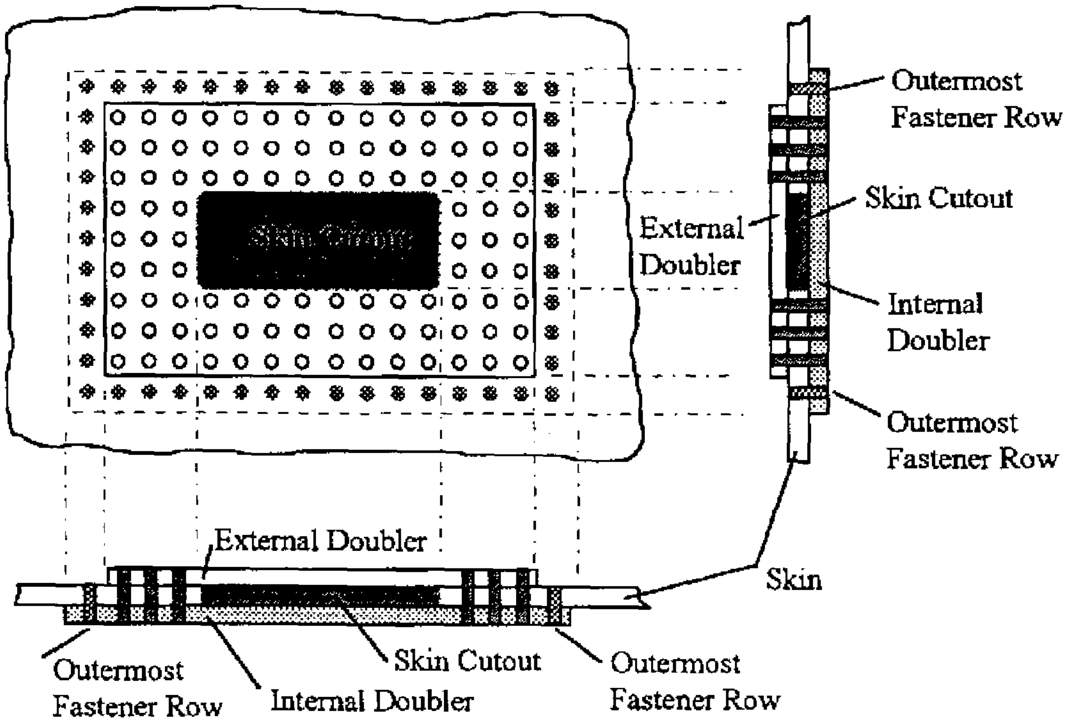

Figure 1. RAPID Software- One Internal and One External Doubler Repair

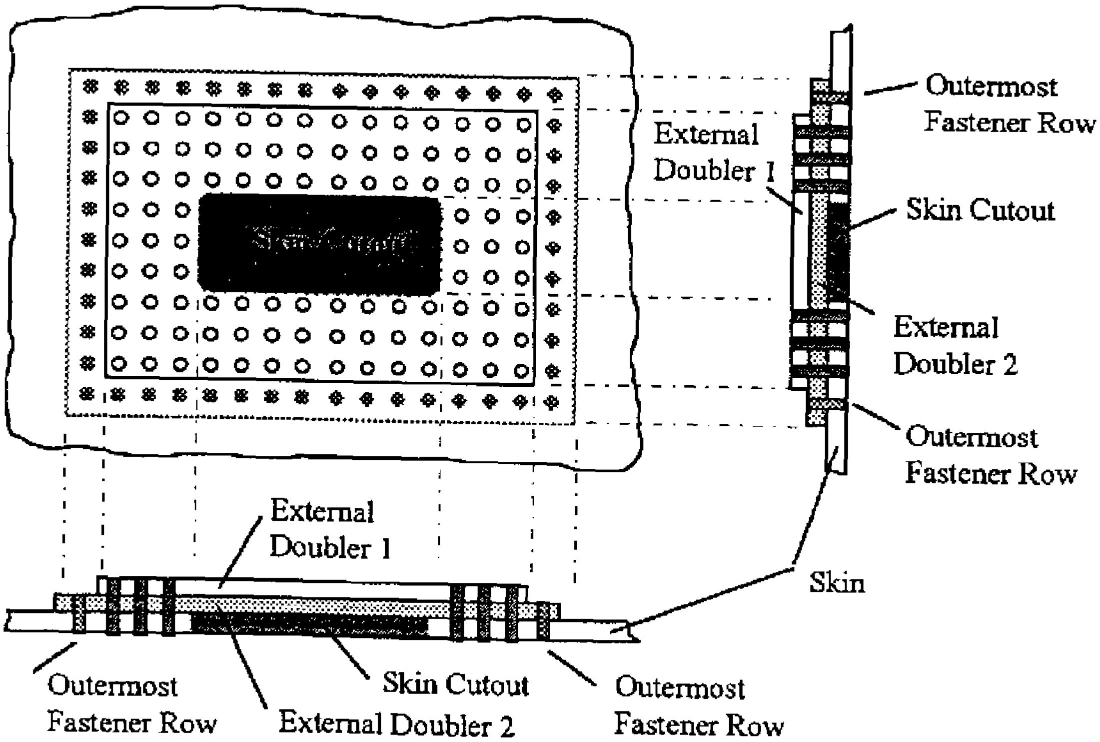

Figure 2. RAPID Software- Two External Doublers Repair

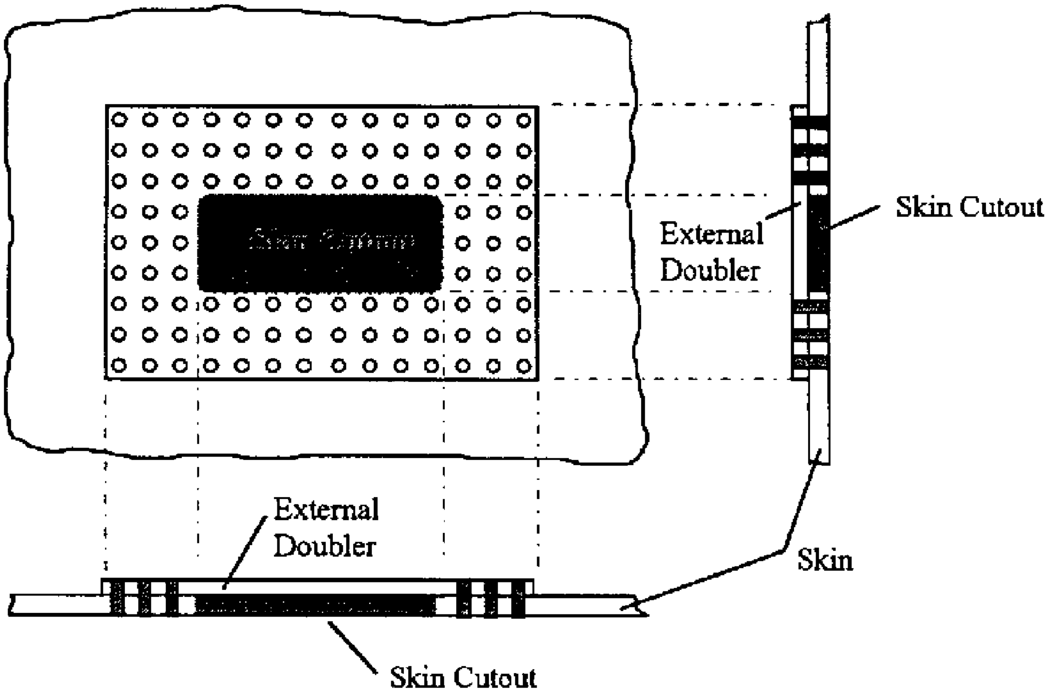

Figure 3. RAPID Software- One External Doubler Repair

4) RAPIDC

RAPIDC is a derivative of RAPID program developed under FAA sponsorship. The program has been developed primarily for mechanically fastened repairs of commuter aircraft. The program is still being beta tested. The program is available in public domain.

5) CalcuRep

This code was developed by Dr. Rob Fredell during his stay at US Air Force Academy in Colorado (Reference 2). This code is for designing bonded repairs for fuselage type of structures subjected to internal pressure loads. The code is available in public domain and is user friendly. The code is primarily useful for designing repairs using Glare material.

6) FRANC2D

This is a finite element code and can be used for crack growth analysis of metallic structures (Reference 3) under constant amplitude loading. The code is available in public domain. Finite element analysis of the structural configuration with crack is carried out and stresses intensity factors determined. The code uses these stress intensity factors for crack growth predictions. A metallic structure with composite patch can be modeled with the code as 3 layers (metal, adhesive and composite patch) and stress intensity factors obtained. These stress intensity factors are used to make crack growth predictions in repaired structure under constant amplitude loading.

3.0 SAMPLE PROBLEMS

3.1 Fuselage Frame Repair

Standard repairs are generally given in repair manuals. However, in many cases in-service inspections show damage that is not covered by standard repair manuals and a special repair has to be designed. For such cases detailed static and damage tolerance analyses have to be carried out. An example of cracked frame in a transport aircraft (Figure 4) is shown in Figure 5. The flange and the web of the frame are cracked as shown in Figure 6a. Standard repair manuals generally do not cover a repair for the damage shown in Figure 5. The cross-sections of the flange and web repairs are shown in Figure 6b. The details of the frame repair are shown in Figure 7.

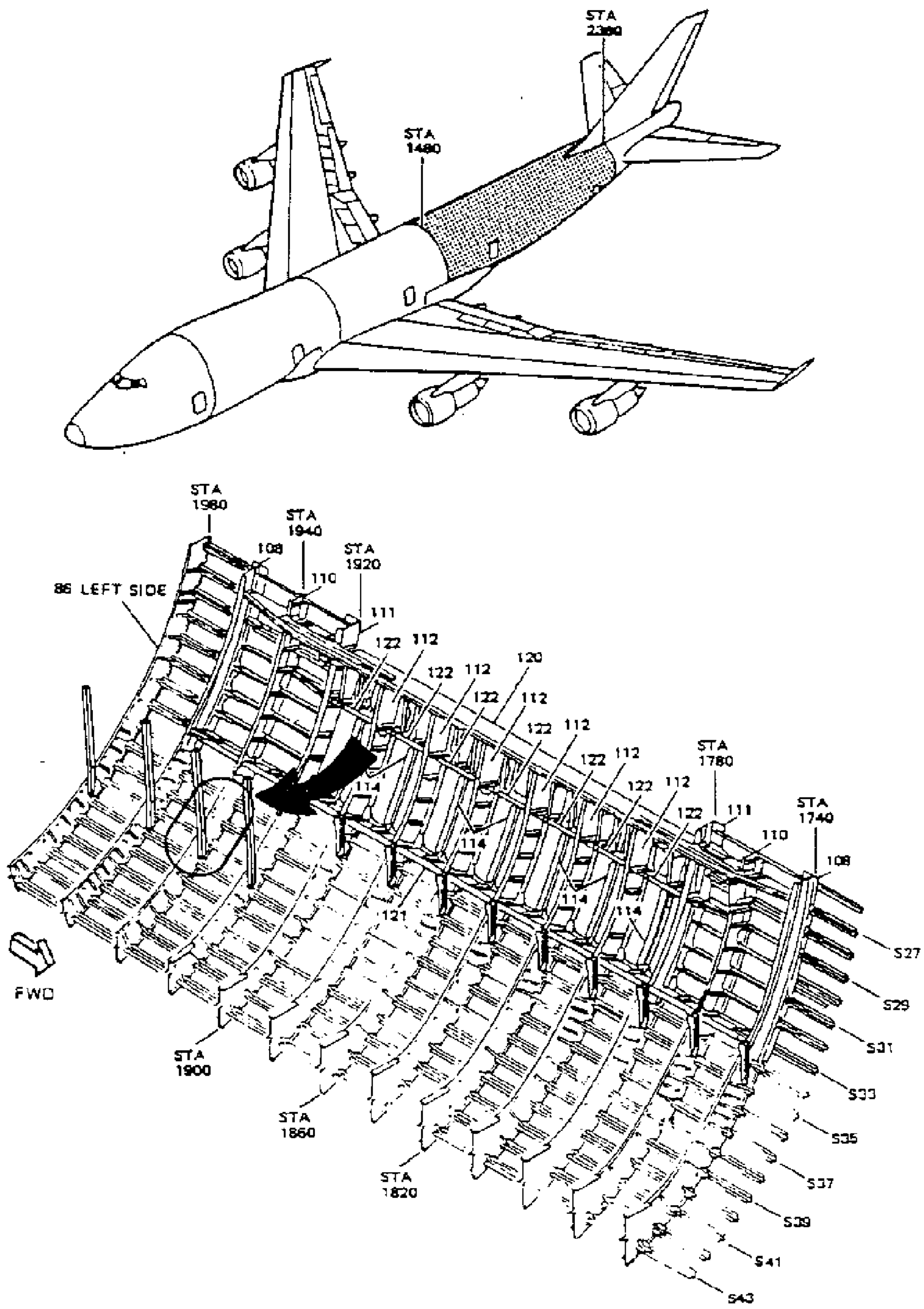

Figure 4. Fuselage Frame Cracking Location in Transport Aircraft

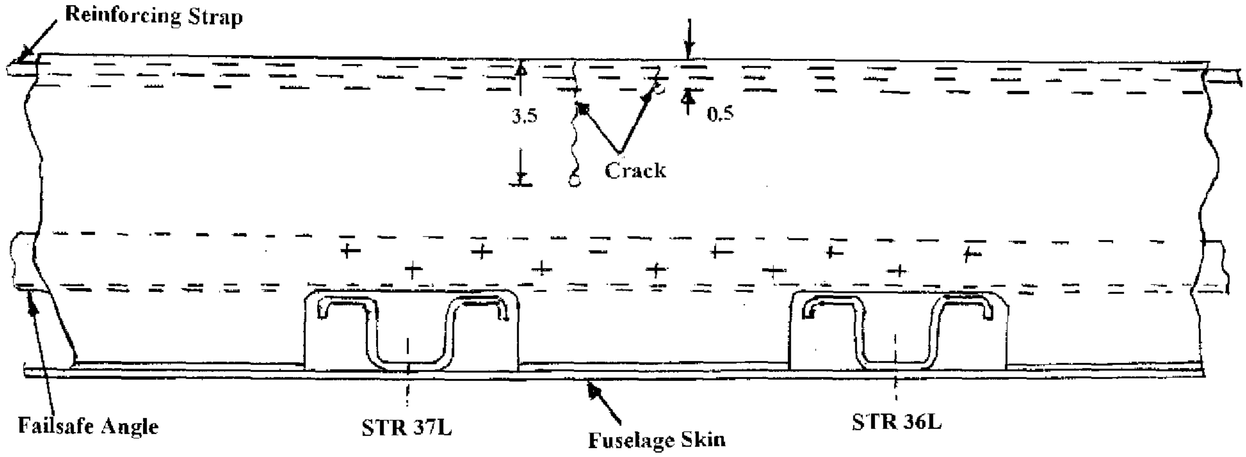

Figure 5. Cracked fuselage Frame

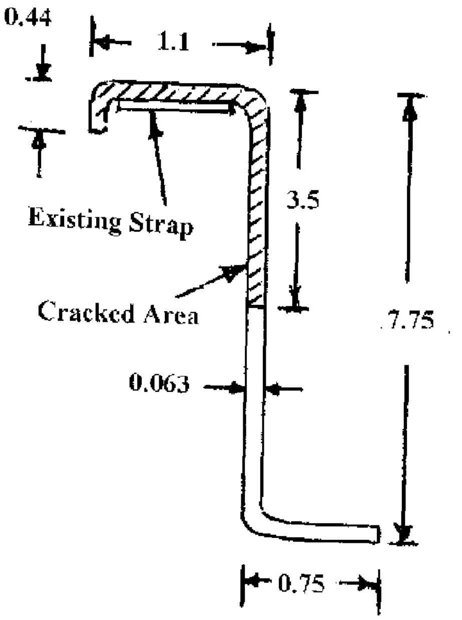

Figure 6a. Section Showing Cracked Frame

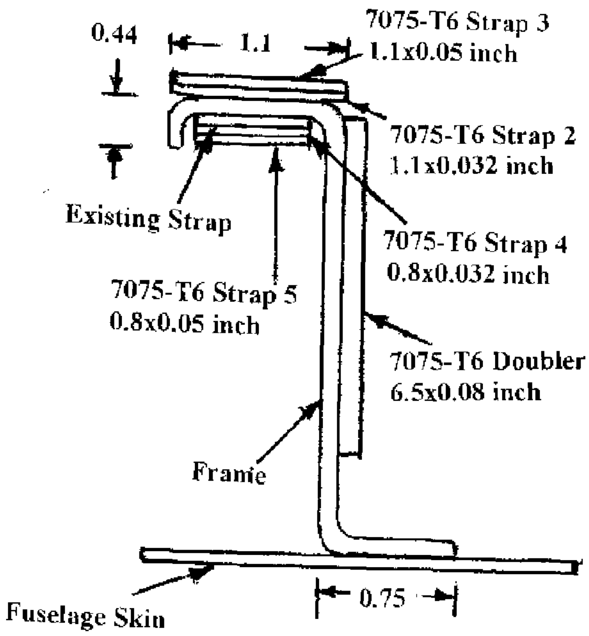

Figure 6b. Section Showing Flange and Web Repairs

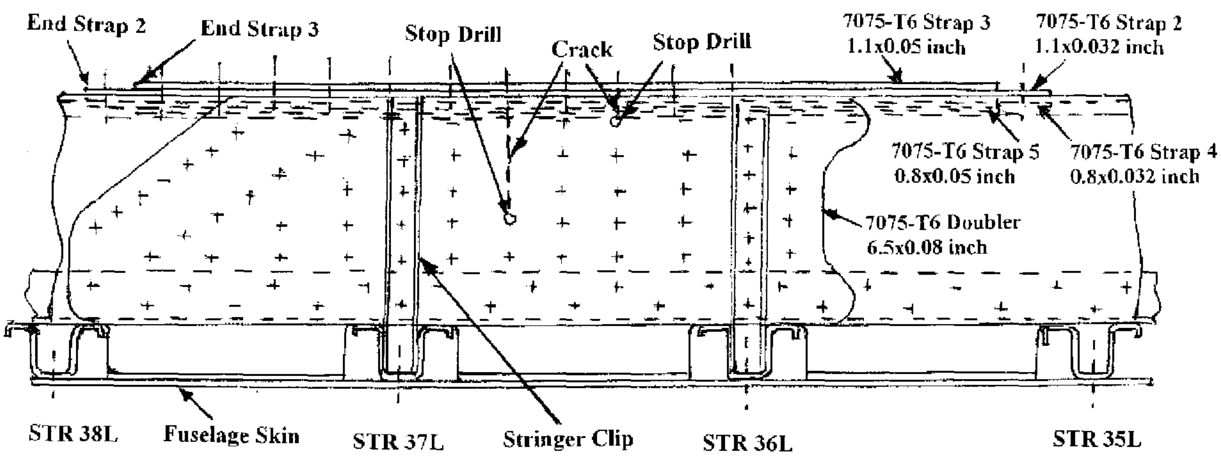

Figure 7. Details of Frame Repair

STATIC STRENGTH DESIGN

Ultimate strength of flange material 7075-T6 aluminum is assumed to be 75 ksi (517.1 MPa).

Load Capacity Lost Due to Cracking

1. Flange area lost due to cracking = (0.44+1.1) x 0.063 = 0.097 in^2 (62.6 mm^2).
Load capacity lost = 0.097 x 75 = 7.275 kips or 7,275 Lb.
2. Web area lost = 3.5 x 0.063 = 0.221 in^2 (142.6 mm^2)
Load capacity lost = 0.221 x 75 = 16.575 kips or 16,575 Lb.
Total Area Lost = 0.097+0.221 = 0.318 in^2 (205.2 mm^2)
Total Load Capacity Lost = 7.275 + 16.575 = 23.850 kips or 23,850 Lb.

Repair Analysis

Repair area required in flange = 1.25 x Area lost = 1.25 x 0.097 = 0.121 in^2 (78.1 mm^2).
Repair area added in flange (Figure 6b) = (0.032 + 0.05) x 1.1 + (0.032 + 0.05) x 0.8
= 0.09 + 0.067 = 0.157 in^2 (101.3 mm^2).

Repair area required in web = 1.25 x 0.221 = 0.276 in^2 (178.1 mm^2).
Repair area added in web = 0.08 x 3.5 = 0.28 in^2 (180.6 mm^2).
Total area added in flange and web = 0.157 + 0.28 = 0.437 in^2 (281.9 mm^2).

Margin of Safety = (Area added / Area lost) – 1.
= (0.437/0.318) - 1. = 1.37 –1. = 0.37

Fastener Requirements

Using HL18-6 HI-Lok in 0.063, 7075-T6 sheet. Allowable loads are given by-
P_S (Shear) = 2,694 Lb.
P_B (Bearing) = 1,197 Lb.
Number of Fasteners Required in Flange = 7,275 / 1,197 = 6 .07.
Use 6 Fasteners.
Number of Fasteners Required in Web = 16,575 /1,197 = 13.8
Use 15 Fasteners.
Total Fastener Load Capacity = 21 x 1,197 = 25,137 Lb.

3.2Composite Reinforcement of T-38 Lower Skin in Machined Pockets

Lower wing skin pockets in T-38 aircraft between the 39% and 44% spars and 33% and 39% spars at Wing Station (WS) 78 have shown a propensity for crack initiation and propagation during service. The cracks have initiated at the pocket radius in the inner moldline of the wing skin. This cracking has been occurring primarily under Lead-in-Fighter (LIF) spectrum loading. These areas are ideal for composite reinforcement to reduce stress levels and enhance fatigue life. As there is no access for bonding reinforcement on the inner moldline, a one sided reinforcement bonded onto the outer moldline of the wing skin was selected (Reference 4). Composite reinforcement bonded to the wing is shown in Figure 8.

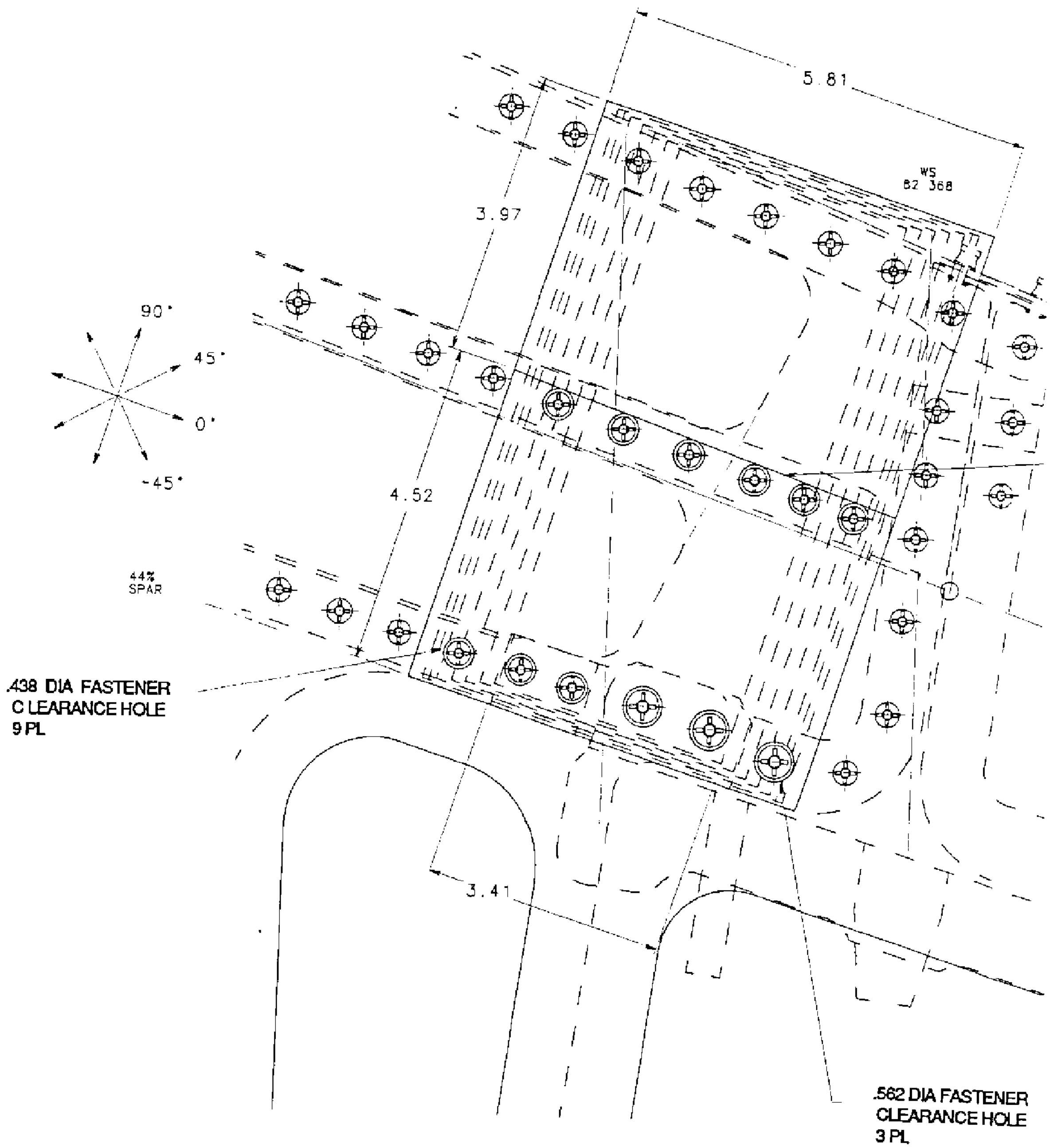

Figure 8. Location of Composite Reinforcement on Lower Wing skin

A detailed finite element analysis of the local area with and without composite reinforcement was carried out (Reference 4). The finite element model of the structure is shown in Figure 9. Typical output of the outer moldline stresses is shown in Figure 10. Using NASTRAN stresses, a detailed damage tolerance analysis of the pocket area was carried out using AFGROW computer code.

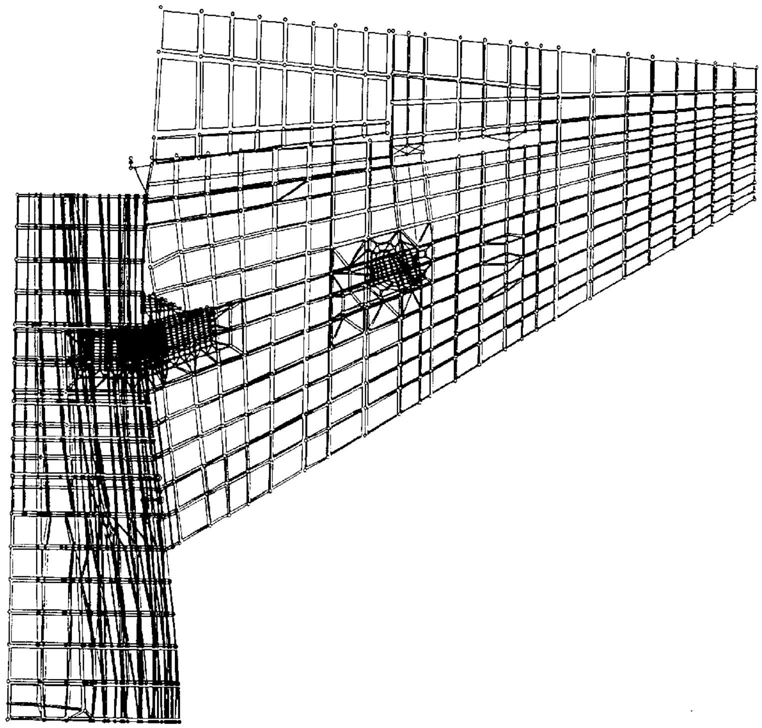

Figure 9. Finite Element Model

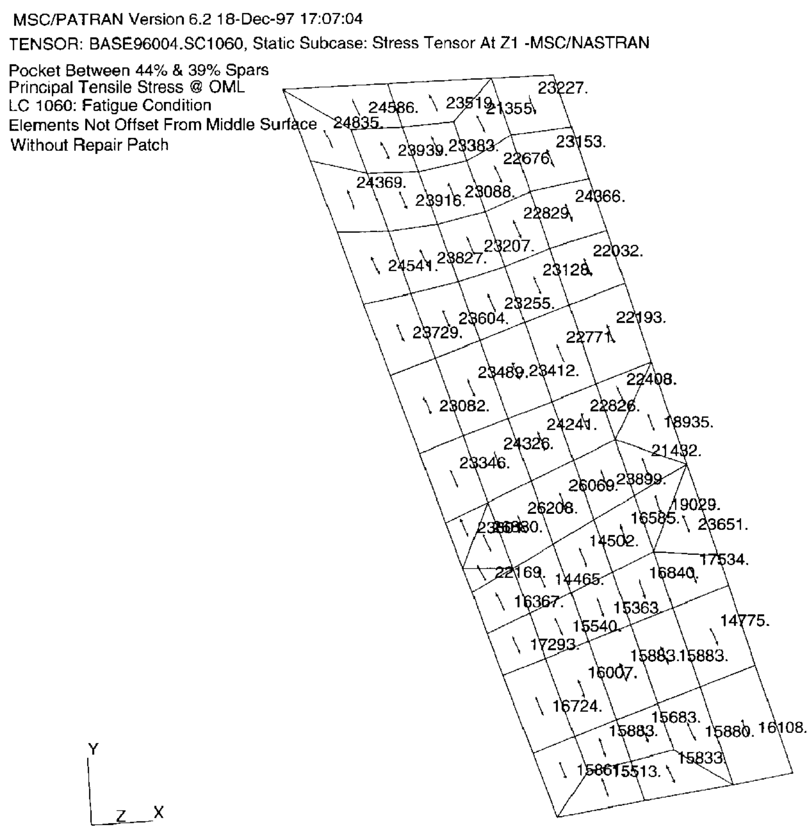

Figure 10. Typical NASTRAN Output Showing Element Stresses

Structural testing of the T-38 wing at Wright Patterson Air Force Base has been carried out under LIF spectrum loading (Reference 4). Prior to the fatigue testing of the wing, strain gages were applied to the lower wing skin pocket area. The strain surveys of the pocket areas were performed prior and subsequent to the bonding of the reinforcement.

The location of the strain gages on the wing is shown in Figure 11. The strain survey was carried out for the two most critical load conditions (namely S0985 and 10556) to 50% of limit load. The stress analysis of the pocket area was performed using coarse mesh and fine mesh finite element models. The comparison of test strains and analytical strains for the pocket between 39% and 44% spars obtained from both models is shown in Figure 12. The figure shows that for condition S0985, both finite element models show a good correlation with the test results. However, for condition 10556, only the fine mesh results correlate well. The coarse mesh results are shown to be slightly higher.

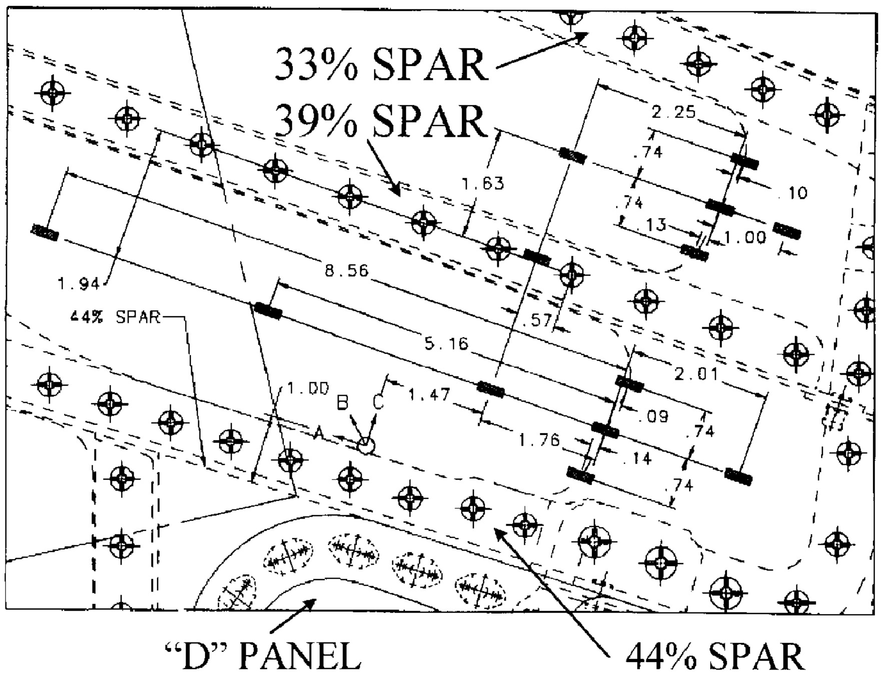

Figure 11. Strain Gage Location in the pocket Area of T-38 Wing Skin

A good comparison between analysis and full scale test indicates that finite element analysis can reliably predict structural behavior with proper modeling techniques and applied boundary conditions. The stresses from finite element analysis along with appropriate loads spectrum can be used in any of the damage tolerance codes to make crack growth and residual strength predictions.

4.0 CONCLUDING REMARKS

A number of codes are available to perform damage tolerance analysis and design repairs for aging aircraft fleet. Most of these codes are available in public domain at no cost to the users. These codes have a good user interface and are user friendly. The operation of these codes does not require expert knowledge. With little training and some knowledge of fracture mechanics, these codes can be very effectively used to make life predictions of aircraft structures. These codes can assist in selecting right life enhancement techniques, designing repairs, and identifying inspection requirements. The application of these codes will reduce down time of an aircraft for repairs.

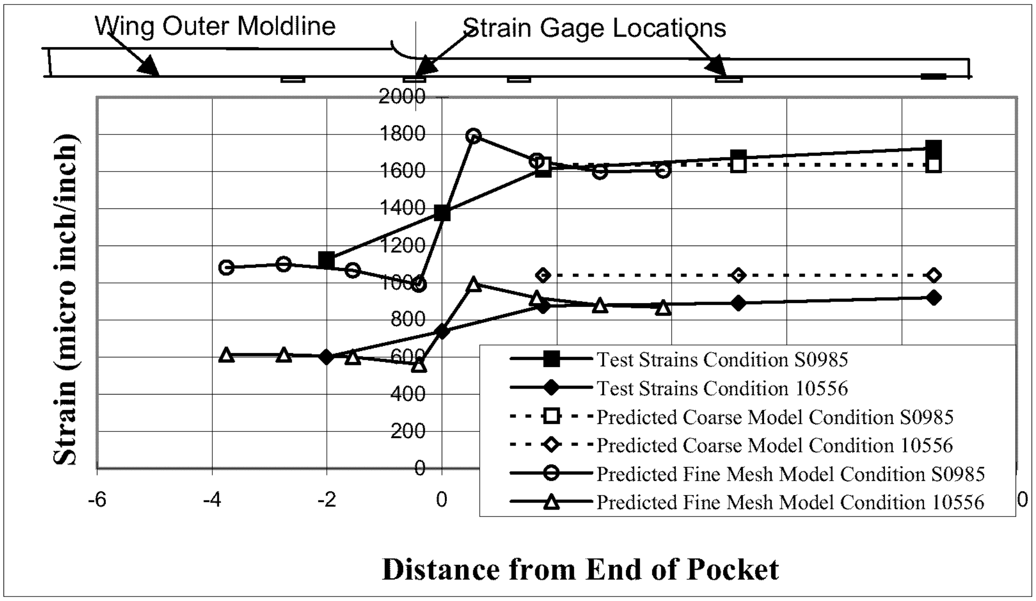

Figure 12. Comparison of Observed and Predicted Strains in Pocket Between 39% and 44% Spars (No Composite Reinforcement)

5.0 REFERENCES

1. Repair Assessment Procedures and Integrated Design (RAPID) User's Manual, 1996.
2. CalcuRep for Windows- User's Manual, Prepared by United States Air Force Academy, may 1995.
3. FRANC2D: A Two-Dimensional Crack Propagation Simulator- User's Manual, NASA Contractor Report 4572, march 1994.
4. Helbling J, Grover R and Ratwani M.M "Analysis and Structural Test of Composite Reinforcement to Extend the Life of T-38 Lower Wing Skin", Proceedings Aircraft Structural Integrity Conference, San Antonio, 1998.